NACA

RESEARCH MEMORANDUM

THE PROSPECTS FOR LAMINAR FLOW ON HYPERSONIC AIRPLANES

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS
WASHINGTON
June 27, 1958

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Ordinarily it is desired to obtain a maximum amount of laminar flow on airplanes for hypersonic flight in order to cut down the aerodynamic heat input and improve aerodynamic efficiency. The location of boundary-layer transition on a complete airplane is affected by a number of factors, some of which are listed in figure 1. There is illustrated an airplane which has the highly swept, blunt-leading-edged airfoil which is required for aerodynamic efficiency and control of the leading-edge temperatures. The first factor listed is the Reynolds number which is well known to be the most important of all. If the Reynolds number is exceptionally low, then laminar flow will occur without regard for the other factors and, conversely, if the Reynolds number is exceptionally high, then fully tur-bulent flow will approximate what is obtained. In the intermediate region the extent of laminar flow depends on other factors, such as the ratio of wall temperature to boundary-layer recovery temperature, the surface rough-ness, the angle of attack, the angle of leading-edge sweepback, and aero-dynamic interference. The configuration of this figure has been selected to emphasize certain kinds of unfavorable aerodynamic interference. We note the possibility of discharge of a turbulent wake by the canard onto the surface of the wing; the crossing of the wing surface by the shock wave generated by the vertical tail; and the intersection of the leading-edge shock waves of all lifting surfaces onto the body boundary layer. All of these interactions might be expected to cause transition to occur at flight Reynolds numbers. A more general type of aerodynamic interfer-ence is the influence on the wing boundary layer of the pressure distribu-tion generated by the body, and vice versa.

Undoubtedly, other factors could be added to this list, such as pressure gradient, important parts of which are included in the category of aerodynamic interference. The point is, however, that all of these factors must be considered if a rational attempt is to be made to maximize the laminar flow, or to predict the extent of the laminar flow.

Examination of the flight Reynolds number shows that it is largely determined by the wing loading and the flight speed as indicated by the equation in figure 2, which, by equating the aircraft weight to the sum of lift and centrifugal force due to the curvature of flight around the earth, defines the air density at equilibrium flight altitude and hence the Reynolds number. A low value of the wing loading permits a high

equilibrium altitude, and consequently a low Reynolds number. Similarly, as the speed becomes appreciable compared to satellite velocity, less aerodynamic lift is required and the altitude is again permitted to increase. The result is that the flight Reynolds number decreases with increase in flight speed. For a wing loading of 25 pounds per square foot and an airplane 50 feet long, the Reynolds numbers based on length are moderate, ranging from 15 million at a Mach number of 4 to 7 million at a Mach number of 20. This indicates that full-scale hypersonic flight can occur in a region of Reynolds number where laminar flow has been observed in flight tests and in wind tunnels. It should be noted, however, that by doubling the wing loading and the length, we will arrive at length Reynolds numbers four times as great as those shown, and therefore considerably less attractive.

The smoothness requirement for hypersonic airplanes is one of the first questions to come to mind. Subsonic experience was that the surfaces had to be exceedingly smooth to permit much laminar flow at flight conditions. The effect on allowable roughness of increasing the flight speed through the supersonic region and into the moderately hypersonic region is shown in figure 3, as it is given by the available test data. Data from references 1, 2, and 3 on three-dimensional roughness, such as sandpaper or carborundum grains on the surface or sandblasting of the surface, are shown in figure 3(a) and are correlated using the Reynolds number based on roughness height and local properties within the boundary layer at the top of the roughness elements. In figure 3(b), data from reference 4 for distributed two-dimensional roughness in the form of transverse grooves are given in terms of a roughness parameter which is primarily the Reynolds number based on roughness height and air properties outside the boundary layer. (The parameter $C = (\delta/x)\sqrt{R}$ is a constant on the surface of a flat plate but varies with Mach number and wall temperature ratio. It expresses the thickness growth rate characteristic of the boundary layer.) The roughness parameter R_h/C is related to the ratio of roughness height to boundary-layer thickness for the case of distributed roughness as has been discussed in reference 5. The trend common to both the two- and three-dimensional roughness data is that the permissible roughness increases with increasing boundary-layer edge Mach number.

Since part (a) of this figure is based on local properties inside the boundary layer, for a constant external flow Reynolds number there is a further effect of Mach number in the relation between local properties at the roughness peak and the external flow properties. The ratio between Reynolds number based on roughness height and external flow properties, R_e, and that based on roughness height and air properties at the roughness peak, R_k, is

$$\frac{R_e}{R_k} = \left(\frac{u_e}{u_k}\right)\left(\frac{T_k}{T_e}\right)^{1.7}$$

As the Mach number increases, T_k increases and, as a result, for a given external flow Reynolds number, the permissible three-dimensional roughness increases with Mach number at a rate faster than indicated in the figure. As an example, the permissible heights of distributed roughness at a Mach number of 7 for the flight conditions of figure 2 are predicted to be greater than 0.2 inch for both two- and three-dimensional roughness.

Another factor which can lead to early occurrence of transition, at least for the case of slender bodies of revolution, is angle of attack. Figure 4, which is taken from reference 6, shows data for the sheltered side of a slender body at angle of attack, the windward side remaining laminar in this case over the entire range. The position at which transition occurred is plotted against the angle of attack in degrees. As the angle goes above about 0.5°, transition comes onto the body at about 27 calibers from the nose and moves forward progressively with increasing angle. This effect is probably a result of transverse pressure gradients on the body causing the boundary-layer profiles to become three-dimensional, a situation similar to that which occurs on sweptback wings. The curves shown on the figure were obtained from the assumption (see ref. 6) that transition occurs when the streamline which crosses the nose-cylinder junction at the 90° meridian reaches a fixed angular position on the sheltered side of the cylindrical cross section, and if assumed angular positions 120° and 180° are taken, the observed points are bracketed. These curves provide a basis for extrapolating the data to higher angles of attack. Of course, the presence of wings on the body would tend to modify the effect and might, in fact, help to suppress it, since the wings act like boundary-layer fences and tend to prevent the body crossflow which leads to transition. At any rate, the adverse effect of angle of attack on bodies should be held in mind.

Sweepback of the wing leading edge has been known to be adverse in subsonic flow since 1952, when flight tests and wind-tunnel tests in England (refs. 7 and 8) brought this to light. Subsonic data, as yet unpublished, recently obtained by Boltz, Kenyon, and Allen in the Ames 12-foot wind tunnel also show this adverse effect and are reproduced in figure 5. Whenever early transition due to sweep has been observed, there has also been observed on the surface a number of parallel lines or streaks with streamwise direction, made visible by the use of subliming agents on the surface as in figure 6. This is a view of the bottom surface of the airfoil in the 12-foot wind tunnel with the flow direction from left to right. The streaks develop in the laminar region and lead individually to separate wedges of turbulence. The white spanwise stripe occurring in the picture was painted on the surface to indicate the chordwise station and was rubbed smooth. It does not influence this result as has been proved by many observations of the phenomenon in which these marks were absent. It was suspected very early that the streaks were traces of streamwise vortices in the boundary layer, and a rake survey of the transverse flow components in the boundary layer made by Boltz confirms this.

It was first suggested by Owen and Randall (ref. 9), that the instability causing transition was due to the three-dimensionality of the boundary layer. Velocities in the transverse direction are induced by transverse pressure gradients present on the surface of even two-dimensional swept wings, because of spanwise "shearing" of the chordwise pressure distribution. Owen and Randall noted that the transverse velocity profile contains an inflection point near its outer edge, and would therefore be unstable. The instability leads initially to streamwise vortices and finally to turbulence. This hypothesis was given further weight by an analysis made by J. T. Stuart (ref. 8), who considered the stability of a boundary layer with three-dimensional profiles when subjected to transverse periodic disturbances, and found that instability exists above a critical Reynolds number. It was found by Owen and Randall that the onset of vortices on the subsonic airfoils occurred when the Reynolds number based on the maximum value of the transverse velocity component and the boundary-layer thickness exceeded a critical value of 125. For the particular airfoils which they considered in subsonic flow, this Reynolds number was found to vary approximately with the square root of the chord Reynolds number and the square root of the nose radius. At values of 130 to 190, transition occurred near the leading edge. In the 12-foot wind-tunnel tests, transition near the leading edge occurs at a value of about 170, in agreement with the British results.

From the explanation of Owen and Randall, it might be expected that sweep would continue to destabilize the boundary layer at supersonic speeds. There is, however, no indication from presently available theory as to what effects compressibility and heat transfer of supersonic boundary layers might have on the stability to this kind of disturbance. The data which are shown in figure 7, however, indicate generally similar end results to those of the subsonic tests. Now it might be expected that a way of combating this instability is to choose wings of constant surface pressure, so that no transverse pressure gradients exist. This kind of pressure distribution is obtained on triangular wings with wedge sections and sharp leading edges when the flow component normal to the leading edge is supersonic. Tests at this condition were made by Dunning and Ulmann, reference 10, at a Mach number of 4 and are indicated in figure 7 by the circles. Surprisingly, an adverse effect of sweepback was still obtained. This might be due to the finite thickness of the leading edge which is necessarily present in experimental models, so that in the vicinity of the leading edge, the flow is three-dimensional; or, it might be a result of some other factors present in the tests of an as-yet unknown character. When the wing section in these tests was changed to a subsonic wing section, so that pressure-distribution effects were introduced, the effect of sweepback became more pronounced, consistent with the hypothesis of Owen and Randall. A test on a triangular wing with 74° sweepback, as yet unpublished, was made in the supersonic free-flight wind tunnel, again using a relatively flat wing section. The leading edge was made blunt intentionally to simulate a leading-edge thickness of 1.8 inches at a full-scale length of 50 feet. The result obtained, when compared with an

earlier data point for a body of revolution, shows a somewhat similar
trend to that obtained on the wedge section wings in the wind-tunnel test,
except at a higher level of Reynolds number. This increase in transition
Reynolds number can be attributed to the reduced wall-temperature ratio
and the reduction in stream turbulence for the flight test.

The effect of a cylindrical leading edge on the transition due to
sweepback has been investigated by Beckwith (ref. 11 and recent unpublished
data). In tests of yawed circular cylinders, he obtained the results indi-
cated by the diamond symbols in figure 7. The Reynolds number for transi-
tion at the stagnation line has been reduced from over 4 million at zero
sweep to the order of 200,000 at 40° of sweepback, based on free-stream
properties and diameter. (Although the data do not indicate definitely
the level of the curve at the higher angles of sweepback, Feller had
reported earlier from tests at a Mach number of 6.9 (ref. 12) that the
cylinder flow was fully laminar at a Reynolds number of 130,000.) For
the flight Reynolds numbers shown in figure 2, these results would
imply turbulent flow only for leading edges larger than about 1-foot diam-
eter. The instabilities generated on smaller leading edges, while not
causing transition on the leading edge, might cause early transition back
on the wing. This possibility will require further investigation.

Owen and Randall's transition criterion has not been evaluated for
the case of the above supersonic data. To do so requires lengthy calcula-
tions of the twisted velocity profiles. In one supersonic experiment for
a Mach number of 1.6 (ref. 13), the transverse critical Reynolds number
was found to be smaller than at subsonic speeds. The promising correla-
tion obtained subsonically makes it appear well worth while to make further
calculations of this kind for supersonic experiments.

Now if we return to the consideration of complete configurations,
and select one which offers some opportunity for maximizing the laminar
flow, it might look something like the three-wing design (ref. 14) which
is shown in the corner of figure 8. This arrangement, by having the wing,
vertical tail, and fuselage originate at a common point, avoids having
shock waves or wakes cross any surface. In addition, the full length wing
panels might be expected to suppress crossflow effects and transition due
to angle of attack. The wing leading edges are swept back 74°, and the
wing surfaces are flat to avoid transverse pressure gradients. However,
the pressure field generated by the body nose induces a transverse pressure
variation onto the surface of the wings, and the leading edge of the wing
is extremely blunt, corresponding to a thickness of 3 inches at a full-
scale length of 50 feet, thereby introducing three-dimensionality into the
boundary layer. (The high degree of bluntness was required to permit gun-
launching the models without incurring buckling failure of the leading
edge.)

Models of this design were tested in the Ames supersonic free-flight
wind tunnel at a Mach number of 6, with a ratio of wall temperature to

recovery temperature of 0.25. Since the boundary-layer condition could
not be visually observed by any available technique, it was decided to
obtain information on the amount of laminar flow from drag measurements.
The data were collected in the manner shown in figure 7, with the mean
square angle of attack as the independent variable in order to correlate
the variations in drag due to lift. In this presentation, the measurements
for constant boundary-layer condition should lie along a straight line
through C_{D_0} with a slope equal to the lift curve slope, and two such
lines are shown. One is for an all-laminar boundary layer and the other
is for an all turbulent boundary layer. The experimental points obtained
may be compared to these two lines.

The models were tested with various degrees of surface roughness, as
noted in the symbol code. The purpose of this was twofold: One class of
surfaces, indicated by the filled symbols, was made very rough in order
to generate an all turbulent boundary layer to establish the accuracy of
the theoretical drag curve. These models had scratches 600 microinches
deep, corresponding to a 1/8 inch depth on the 50-foot airplane, and the
scratches were applied particularly across the face of the leading edge,
with the thought that this would be effective in causing immediate transi-
tion. The results obtained confirm the calculation for turbulent boundary
layer. The other models, the open symbols, had surface roughnesses which
were well below the critical values indicated by figure 3, and the rough-
ness was varied in this region to see if corresponding variations in drag
would be observed. Within the scatter, no dependence of the drag on
smoothness was observed. The roughest of these surfaces would correspond
to a surface covered with broad scratches 1/32 inch deep at full scale.

The open symbols, with the exception of one point, define a line
roughly parallel to the theoretical lines, at a level about 1/4 to 1/3 of
the way between the laminar and turbulent lines, which indicates that the
boundary layer is laminar over 2/3 to 3/4 of the model surface. Figure 7
shows for the case of a 74° swept wing alone in the same test facility,
a transition Reynolds number of 3.3 million. If transition is assumed to
occur on the airplane model at this value of the Reynolds number at all
spanwise stations, there results a triangular area of turbulent flow of
about 1/2 the length of the model, and therefore of about 1/4 the wetted
area leaving 3/4 of the wetted area laminar. The present result with the
airplane model is therefore consistent with the result for the wing alone
from the earlier figure. This correspondence of the two results would
imply that there are no seriously unfavorable effects of configuration on
the airplane as tested, compared to the wing alone. It also appears that
in both cases, sweepback was the predominant factor leading to transition.

At higher Reynolds numbers, the results indicated in figure 9 were
obtained. As would be expected, at Reynolds numbers between 12 and
13 million, the measured drag was closer to the turbulent curve, corre-
sponding to a laminar boundary layer on perhaps 45 percent of the surface.
This result continues to correspond to transition at a streamwise Reynolds
number of 3.3 million.

In conclusion, some of the factors which must be considered in relation to boundary-layer transition on hypersonic airplanes have been reviewed. It appears that the Reynolds numbers for full-scale flight will be moderate, for vehicles of the size of fighter aircraft, if low wing loadings are employed. Roughness effects give promise of being much less severe than in lower speed flight, which is not to say that roughness can be ignored altogether as a factor at hypersonic speeds. Angle of attack and aerodynamic interference effects are of sufficient importance to warrant attention in designs which seek to obtain the maximum extent of laminar flow. The principal deterrent to fully laminar flow is the adverse effect of sweepback of the wing leading edge. Up to the present time, transition has been observed to occur at Reynolds numbers no higher than 3.3 million on a 74° swept wing with blunt leading edge at a Mach number of 6 under temperature conditions similar to those of flight. This was sufficient to give laminar boundary layer on from 1/2 to 3/4 of the model surface at flight Reynolds numbers (for a wing loading of 25 lb/sq ft^2). The effect on this result of further refinements in design cannot be foretold, as transition due to sweepback is only beginning to be understood. Further changes in result might also be anticipated from increasing the Mach number well above 6 and correspondingly reducing the ratio of wall temperature to recovery temperature. These possibilities must await further study.

Ames Aeronautical Laboratory
    National Advisory Committee for Aeronautics
        Moffett Field, Calif., Mar. 20, 1958

APPENDIX

LIST OF SYMBOLS

C   boundary-layer thickness constant $= \dfrac{\delta}{x}\sqrt{R}$

C_D drag coefficient

C_{D_O} minimum drag coefficient

C_L lift coefficient

d   diameter, ft

l airplane length, ft

M   Mach number

R   Reynolds number based on length and free-stream air properties

R_h Reynolds number based on roughness height and air properties at the boundary-layer edge

R_k Reynolds number based on roughness height and air properties at the roughness peak

S   area of lifting surface, sq ft

T   temperature, ^OR

u   free-stream air velocity, ft/sec

u_s satellite velocity, ft/sec

W   airplane gross weight, lb

x   axial coordinate, ft

α angle of attack, deg

$\overline{\alpha_R^2}$ mean square value of the resultant angle of attack, deg^2

β $\sqrt{M^2 - 1}$

δ boundary-layer thickness, ft

Λ angle of leading-edge sweepback, deg

μ viscosity in free stream, lb sec/sq ft

ρ free-stream air density, slugs/cu ft

Subscripts

e   boundary-layer edge

∞ free stream

k   air properties at the roughness peak

r   recovery or adiabatic wall conditions

T   transition

W   wall or model surface

NACA RM A58D25

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FACTORS AFFECTING EXTENT OF LAMINAR FLOW
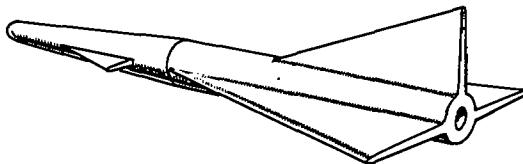
1. REYNOLDS NUMBER
2. WALL TEMP RATIO
3. ROUGHNESS
4. ANGLE OF ATTACK
5. SWEEPBACK
6. AERODYNAMIC
    INTERFERENCE

Figure 1

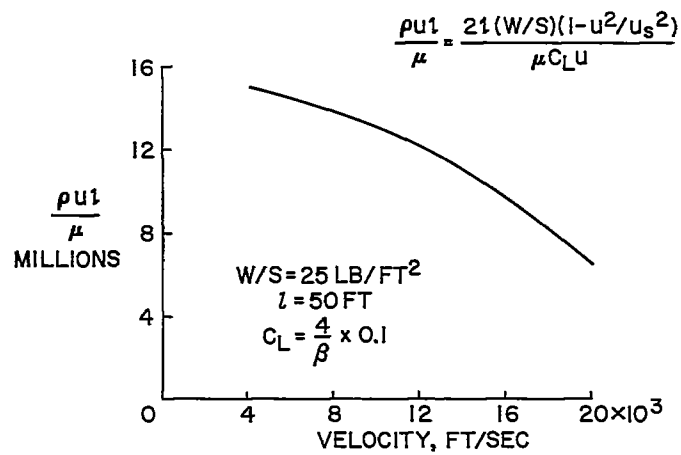FLIGHT REYNOLDS NUMBERS OF HYPERSONIC AIRPLANE

$$\frac{\rho u l}{\mu} = \frac{2l(W/S)(1-u^2/u_s{}^2)}{\mu c_L u}$$

$\dfrac{\rho u l}{\mu}$ MILLIONS

W/S = 25 LB/FT^2
l = 50 FT
$c_L = \dfrac{4}{\beta} \times 0.1$

VELOCITY, FT/SEC

Figure 2

EFFECT OF MACH NO ON ALLOWABLE ROUGHNESS

3-DIMENSIONAL ROUGHNESS
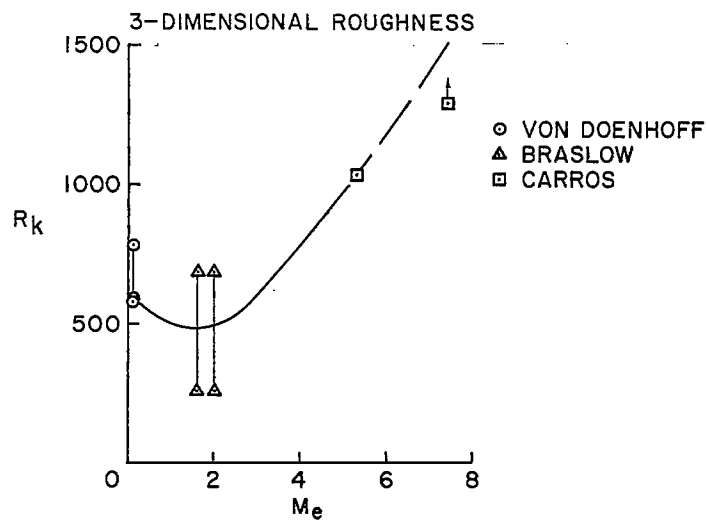
Figure 3(a)

EFFECT OF MACH NO ON ALLOWABLE ROUGHNESS

2 - DIMENSIONAL ROUGHNESS
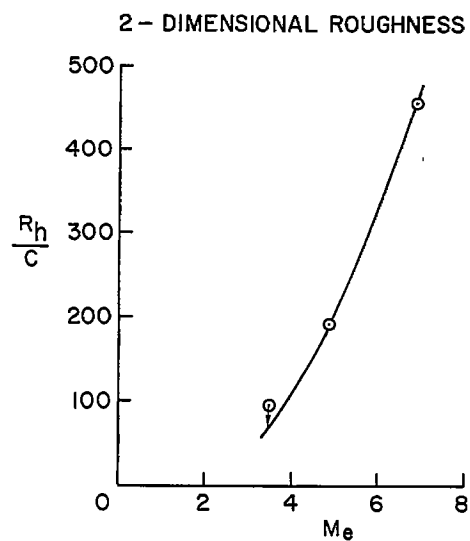
Figure 3(b)

SHELTERED-SIDE TRANSITION ON BODY
AT ANGLE OF ATTACK

$$M = 3.5$$
$$R = 11 \times 10^6$$

$$\theta_T = 120° \quad = 180°$$

$$\frac{x_T}{d} = \frac{x_n}{d} + \frac{\sin(\theta_T - 90)}{2\alpha}$$

α, DEG

$\frac{x_T}{d}$
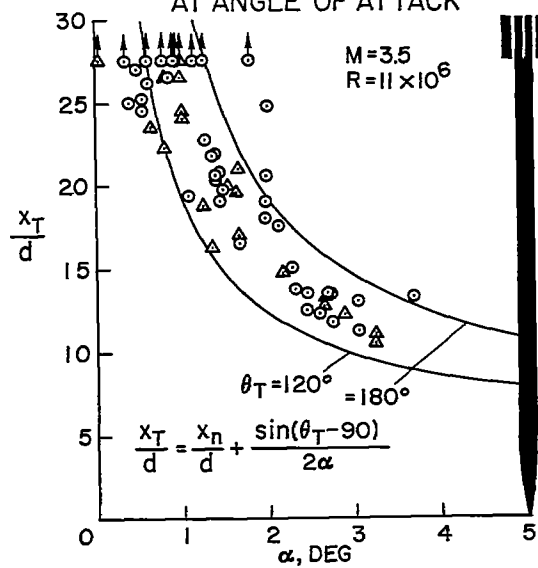
Figure 4

EFFECT OF SWEEPBACK, SUBSONIC DATA

64_2A015 AIRFOIL
AMES 12-FOOT W T

R_T, MILLIONS

$\alpha = 2°$, LOWER SURFACE

$\alpha = 0$

$\alpha = 2°$, UPPER SURFACE

Λ, DEG
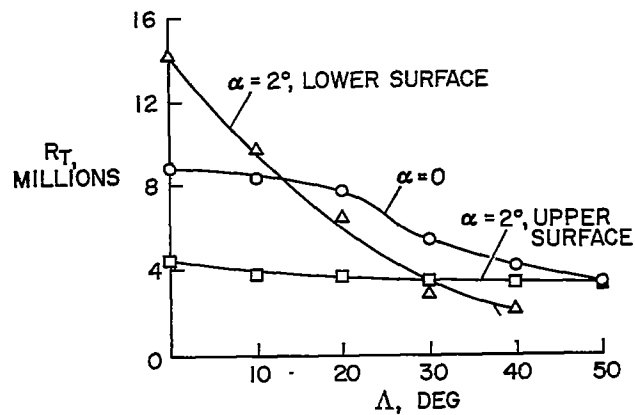
Figure 5

TRANSITION PATTERN ON SUBSONIC AIRFOIL

Figure 6

EFFECT OF SWEEPBACK, SUPERSONIC DATA
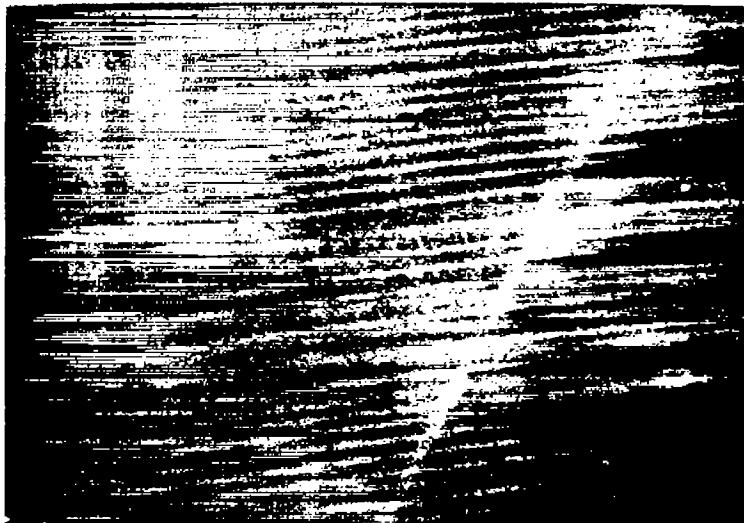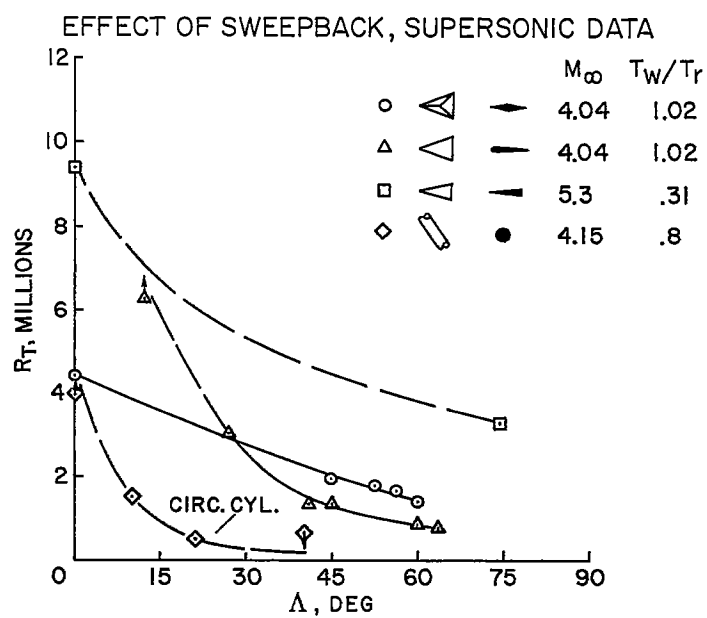
Figure 7

COMPARISON OF DRAG MEASUREMENT WITH
THEORY FOR 3-WING AIRPLANE MODEL

CALCULATED
TOTAL DRAG

TURBULENT

LAMINAR

C_D

M = 6
R = 5.7 TO 7.6 x 10^6

- ○ BUMPS, < 5 μ INCH
- △ SCRATCHES, 30 μ INCH
- □   "   , 150 μ INCH
- ▼   "   , 600 μ INCH
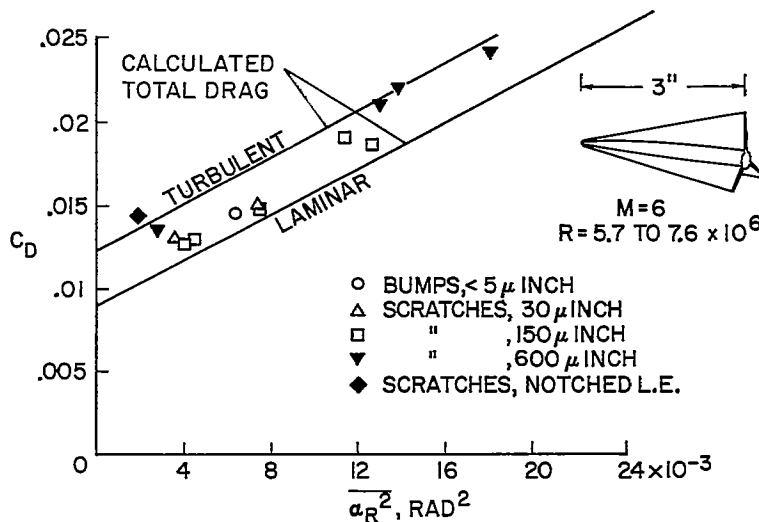- ◆ SCRATCHES, NOTCHED L.E.

$\overline{\alpha_R^2}$, RAD^2

Figure 8

VARIATION OF MINIMUM DRAG WITH REYNOLDS NUMBER

M = 6

C_{D_O}

TURBULENT

LAMINAR

REYNOLDS NUMBER, MILLIONS

Figure 9